



RESEARCH MEMORANDUM

EXPLORATORY INVESTIGATION OF THE MOMENTS ON OSCILLATING
CONTROL SURFACES AT TRANSONIC SPEEDS

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EXPLORATORY INVESTIGATION OF THE MOMENTS ON OSCILLATING

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SUMMARY

Hinge-moment data have been obtained for oscillating control surfaces on swept, unswept, and delta wings through the use of wind-tunnel and rocket models. The in-phase and damping-moment coefficients were measured and a range of unstable aerodynamic damping was found at transonic speeds for each of the configurations tested. The magnitudes of the hinge-moment coefficients are given and, since no systematic theory that would account for separated-flow or aspect-ratio effects was available, comparisons are made with theory based on two-dimensional potential flow for subsonic, sonic, and supersonic speeds. A rather surprising agreement with theory is noted for a range of conditions where the theory would not be expected to apply. Although the theory is inadequate in predicting the magnitudes of the damping coefficient in the transonic speed range, some of the trends seem to be correctly given. The results show the importance of several factors: for example, the dependence of the damping-moment coefficients upon the amplitude of oscillation, initial angle of attack, and reduced frequency. The results indicate that troubles caused by transonic control flutter may be alleviated to some extent by the use of dampers, structural modifications, or by aerodynamic changes.

INTRODUCTION

One of the most difficult problems that has arisen as flight speeds have increased into the transonic and supersonic speed range is concerned with control-surface flutter. Flutter troubles on control surfaces have been the rule rather than the exception on most configurations. Broadly speaking, there are two types of flutter involving control surfaces that have been of concern. One is coupled flutter that involves an interaction between control-surface motion and one or more other degrees of freedom of the airplane. Adjusting the mass balance, for example, as directed by theory has usually proved adequate to correct this coupled control-surface flutter. However, even though the coupled flutter is eliminated, another type of flutter involving only the degree of freedom of the control surface is frequently encountered. (See refs. 1 to 9.) This single-degree-of-freedom control-surface flutter is generally not sensitive to

mass balance. However, like most single-degree-of-freedom types of flutter, it is very sensitive to damping.

To determine the amount of damping necessary to prevent this single-degree type of flutter, a number of experimental measurements have recently been made at transonic speeds of the hinge moments on control surfaces on swept, unswept, and delta wings. Presenting some preliminary results of these investigations is the primary purpose of this paper.

SYMBOLS

| | |
|--|--|
| M_δ | aerodynamic hinge moment on control per unit deflection, positive trailing edge down, ft-lb/radian |
| q | free-stream dynamic pressure, lb/sq ft |
| M' | area moment of aileron area rearward of and about hinge line, ft ³ |
| c_a | mean geometric control chord, ft |
| c_w | mean geometric wing chord, ft |
| ω | angular frequency of oscillation, radians/sec |
| k_a | aileron reduced frequency, $\frac{\omega c_a}{2V}$ |
| k_w | wing reduced frequency, $\frac{\omega c_w}{2V}$ |
| V | free-stream velocity, ft/sec |
| M | Mach number |
| C | equivalent viscous-damping coefficient, $\frac{\text{ft-lb}}{\text{radians/sec}}$ |
| δ | control-surface deflection, positive trailing edge down, radians |
| $1 - x_1$ | ratio of control chord to wing chord |
| M_3, M_4, N_5, N_6 M_3', M_4', N_5', N_6' | } flutter derivatives as used, for example, in reference 10 |

L control span, ft

$$C_{h\delta} = \frac{\text{Real part of } M_\delta}{2M'q}$$

$$C_{h\dot{\delta}} = \frac{\text{Imaginary part of } M_\delta}{2M'qk_a}$$

DISCUSSION OF PARAMETERS

The hinge moment existing on an oscillating control is not necessarily in phase with the control position and may be represented in complex notation by the relation

$$\frac{M_\delta}{2M'q} = C_{h\delta} + ik_a C_{h\dot{\delta}}$$

The part $C_{h\delta}$ is the component in phase with the displacement and is commonly called the inphase or spring moment, whereas $k_a C_{h\dot{\delta}}$ is the component that is 90° out of phase with displacement, that is, in phase with the velocity. This part is called the quadrature or damping moment. Negative values of $C_{h\delta}$ oppose the displacement and hence act as an aerodynamic spring and result in an increase in the stiffness or an increase in the natural frequency of a control surface. Likewise, negative values of $C_{h\dot{\delta}}$ oppose the velocity and hence indicate stable damping; that is, a free oscillation of a control surface would damp out. Positive values of $C_{h\dot{\delta}}$ then would indicate an unstable aerodynamic damping moment, and an oscillation would increase in amplitude unless structural damping or a control-surface damper provided damping moments greater than the unstable aerodynamic moments. The value of equivalent viscous damping required of the damper to overbalance the unstable aerodynamic moment is given by the expression

$$C = \frac{qM'c_a C_{h\dot{\delta}}}{V}$$

where C is the damper hinge moment in foot-pounds per angular velocity required of the damper.

The data presented in this paper are in the form of the stability coefficients $C_{h\delta}$ and $C_{h\dot{\delta}}$; the expressions relating these values to commonly used coefficients in flutter analysis are

$$C_{h\delta} = - \frac{c_a^2 k_a^2 I M_3}{M'} = - \frac{c_a^2 k_w^2 I N_5}{(1 - x_1)^2 M'}$$

and

$$C_{h\dot{\delta}} = - \frac{c_a^2 k_a^2 I M_4}{M'} = - \frac{c_a^2 k_w^2 I N_6}{(1 - x_1)^3 M'}$$

For the special case of a rectangular control hinged at the leading edge,

$$C_{h\delta} = -2k_a^2 M_3' = - \frac{2k_w^2 N_5'}{(1 - x_1)^2}$$

$$C_{h\dot{\delta}} = -2k_a M_4' = - \frac{2k_w N_6'}{(1 - x_1)^3}$$

THEORETICAL CONSIDERATIONS

It may be of interest to see just what the theory predicts for the control-surface damping moments throughout the transonic speed range. Figure 1 shows theoretical values of $C_{h\dot{\delta}}$, the control damping coefficient, as a function of Mach number for three values of reduced frequency based on control chord. These values have been obtained from references 10, 11, and 12 for the subsonic, sonic, and supersonic ranges. All calculations are for a 20-percent-chord control hinged at its leading edge. No two-dimensional coefficients are tabulated between $M = 0.8$ and 1.0, and hence the curves have been arbitrarily faired between the subsonic and sonic theories. Theory shows that for the lower range of reduced frequencies there is an abrupt loss in stable damping and that the damping becomes unstable and remains unstable up to supersonic speeds. Theory further indicates that at the higher reduced frequencies the instability does not exist throughout the speed range. This has been confirmed by experience inasmuch as it has generally been found that, if it is possible to make the control-surface frequency high enough, the troubles have been cured or avoided. When an excessive penalty must be paid to achieve a sufficiently high frequency, it has been necessary to provide dampers to absorb the unstable aerodynamic damping that remains.

DISCUSSION OF RESULTS

Now that the predictions of the idealized theory have been considered, some experimental results in the transonic speed range are discussed. Experimental data are somewhat difficult to correlate because of nonlinearities that are encountered on control surfaces. One nonlinear effect is illustrated in figure 2, which shows the experimental variation of the damping-moment coefficient $C_{h\delta}$ with the amplitude through which the control is oscillating. These data are for an unswept, semi-span model which was tested in the Langley high-speed 7- by 10-foot tunnel at a Reynolds number of about 2×10^6 based on wing chord. The 25-percent-chord aileron had 20-percent aerodynamic balance and was not sealed. It can be seen that at these Mach numbers there is a nonlinear variation of damping-moment coefficient with amplitude. Further, for this particular case the maximum unstable damping appears to occur at some intermediate amplitude, and it is possible that, combined with some level of structural damping, this could explain some of the limited-amplitude flutter obtained in many cases of control-surface flutter. These nonlinear variations with amplitude, however, make evaluations of the effects of other parameters, such as Mach number, difficult.

In order to obtain some idea of Mach number effects, a constant amplitude of oscillation was chosen near the maximum unstable damping, around 2.5° , and the damping coefficients for this amplitude were plotted as a function of Mach number. Figure 3 shows the experimentally determined damping-moment coefficients at angles of attack of 0° and 6° for the same configuration, and the dashed curve indicates the values predicted by two-dimensional subsonic, sonic, and supersonic theories. The theoretical values presented in this figure as well as subsequent figures were calculated for a 20-percent-chord control hinged at the leading edge. It can be seen from the curves through the data points that there is an abrupt change from stable to unstable damping, and it has been found that the Mach number at which this change takes place depends upon many things, for example, airfoil thickness, angle of attack, or amplitude of oscillation.

Of immediate importance to the transonic and supersonic airplane designer is the magnitude of the maximum unstable damping that is likely to be encountered over the entire Mach number range. It can be seen that theory, which is the idealized two-dimensional theory, predicts some of the trends but is inadequate in predicting the magnitudes. The magnitude thus depends upon oscillation amplitude as was seen in figure 2 and, to a lesser extent, angle of attack as indicated in figure 3.

The aerodynamic profile of the control is known to have an effect on aileron buzz, and figure 4 shows the effect of control-surface trailing-edge thickness on the damping-moment coefficient. Control

surfaces with thickened trailing edges have been found in some cases to be less susceptible to control-surface buzz, and the results of this figure show that the control surface with a thickened trailing edge had smaller unstable damping moments than the one with a sharp trailing edge. Maximum values of $C_{h\delta}$ over an amplitude range of $\pm 10^\circ$ at zero angle of attack were used for this comparison.

Also of interest to the airplane designer is the aerodynamic inphase or spring moment, and figure 5 shows the inphase moment coefficient plotted against Mach number for the same two aileron profiles as in figure 4. The coefficient $C_{h\delta}$ is the inphase aerodynamic moment coefficient, and negative values, it may be recalled, indicate a stiffening or spring effect. It is seen that $C_{h\delta}$ is negative throughout the Mach number range, and it is of interest in comparing the effect of the control profile that the magnitudes of the inphase moments are increased when the trailing edge is thickened, whereas the damping moments are decreased; this would indicate a large reduction in the phase angle of the moment vector as the trailing edge is thickened. Theory again follows the general trend but predicts too large a magnitude. However, the theory shown was for a two-dimensional control with a sealed gap and hinged at the leading edge, whereas the control for this experiment permitted flow through the gap and had 20-percent aerodynamic balance.

Swept-wing controls have also encountered control-surface instabilities, and figure 6 presents the damping-moment coefficients on a swept-rudder configuration having a 25-percent-chord control hinged at the leading edge. These data were obtained from tests of a 5-percent-thick semispan model in the Langley 8-foot transonic pressure tunnel. The data are representative of oscillation amplitudes of $\pm 10^\circ$, zero angle of attack, and Reynolds numbers of about 6×10^6 . These tests extended to supersonic speeds of about $M = 1.12$ and again indicate an abrupt loss in damping, as in the case of the unswept configuration. The theory and experiment are for a constant value of reduced frequency of 0.048 and the experimental curve is obtained from cross plots of data. The theory is based on the component flow Mach number perpendicular to the hinge line. Although the trend of the instability seems to be predicted by theory, the crossover points and the magnitudes are in error. The unstable damping region obtained experimentally occurs at a slightly higher Mach number than that for the unswept wing, although, as mentioned, not as high as that predicted by the component flow Mach number theory.

The inphase hinge moments for the same configuration as in figure 6 are shown in figure 7, and a very good if not coincidental agreement is noted with theory.

Although control-surface instabilities on delta wings have not been as documented as those for other types of configurations, experimental hinge-moment measurements have been obtained for oscillating delta-wing controls and the damping-moment components are shown in figure 8. Some data are shown for a full-span model tested in the Ames 6- by 6-foot supersonic tunnel (ref. 13 and unpublished data) for a control with a sharp trailing edge. These data were obtained at zero angle of attack, oscillation amplitudes of $\pm 10^\circ$, a Reynolds number of 2.4×10^6 based on wing mean aerodynamic chord, and reduced frequencies up to 0.03. Additional data are shown for a rocket model launched by the Langley Pilotless Aircraft Research Division at zero angle of attack with a full-span constant-chord unbalanced control having a thickened trailing edge. The reduced frequency for this test varied from 0.09 to 0.03 between Mach numbers of 0.3 and 1.9, and the Reynolds number based on the wing mean aerodynamic chord ranged from 3.5×10^6 to 18×10^6 . The damping-moment coefficients for the delta wing also show a loss in stable damping at transonic speeds, and stable damping appears to be regained at supersonic speeds, depending upon the amplitude of oscillation. The rocket model encountered control-surface flutter in the range of Mach number indicated by the hatched area and appeared to become stable above a Mach number of about 1.3. The control remained stable up to the maximum speed of the flight around $M = 2.0$, although a failure in the oscillating mechanism precluded obtaining damping coefficients in this range. Stiffness coefficients $C_{h\delta}$ were obtained for the same configurations and are shown in figure 9. The measured stiffness coefficients increase as sonic speeds are approached and decrease at supersonic speeds in much the manner that theory predicts. The theory is for a sealed gap, whereas the tunnel experiments permitted some flow.

CONCLUDING REMARKS

The results of the investigations thus far have indicated that the airplane designer has several measures at his disposal for solving the problem of single-degree-of-freedom control-surface flutter. Aerodynamic modifications appear to offer some promise but require considerably more study to establish trends that will be practical for design. Structural modifications that increase the stiffness and, hence, frequency of the control appear to be straightforward, although there are limits to the amount by which the control-surface frequency can be increased before excessive weight penalties or other complications are encountered. The addition of control-surface dampers appears to offer another means of eliminating the control-surface instabilities, and some of the data of this paper may be useful for this purpose. Of course, each basically different configuration will require separate study. It must be pointed out that, if the control-surface frequency is low, the size of the damper required to overcome the large unstable aerodynamic damping encountered

at the low reduced frequencies may impose restrictions on the rate at which the pilot may control the airplane. Hence, it appears that some kind of a compromise may be necessary between control-surface stiffness and damper size.

Langley Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., May 16, 1955.

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THEORETICAL DAMPING COEFF. AS FUNCTION OF MACH NO.
AND REDUCED FREQUENCY

2-DIM. COMP.-FLOW THEORY

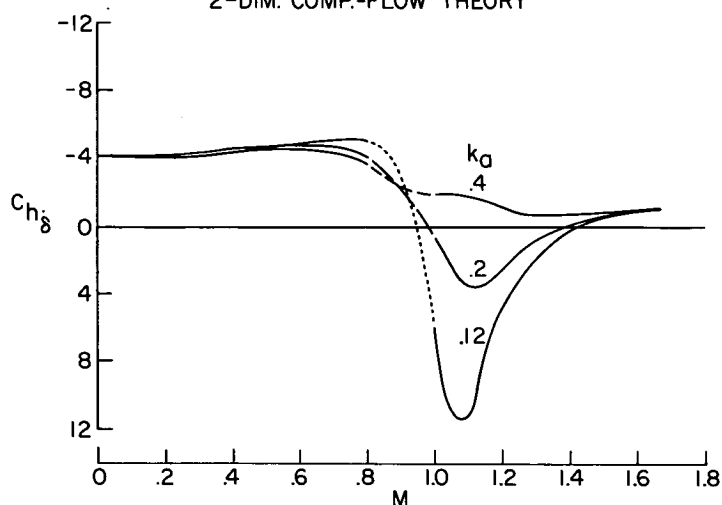


Figure 1

EFFECT OF AMPLITUDE ON CONTROL-SURFACE
DAMPING COEFFICIENT

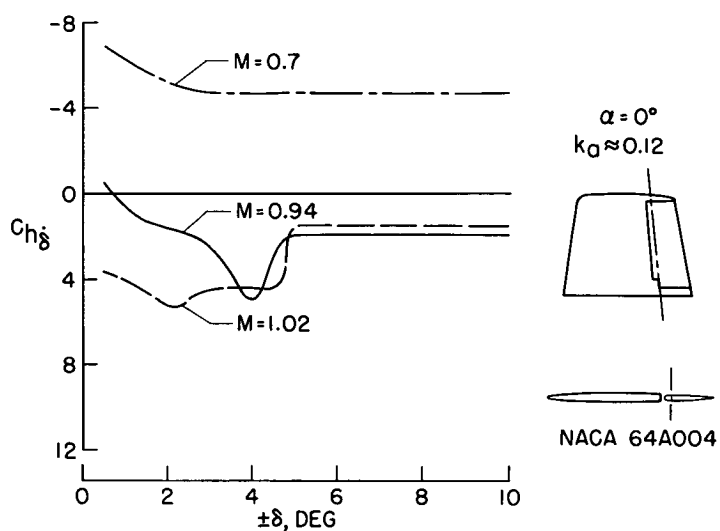


Figure 2

DAMPING COEFFICIENT FOR UNSWEPT-WING CONTROL
 $k_a \approx 0.12$

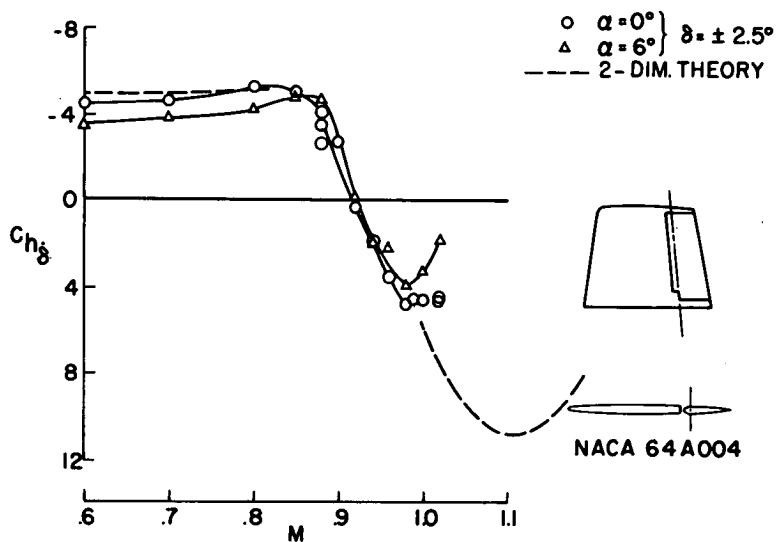


Figure 3

DAMPING COEFFICIENT FOR UNSWEPT-WING CONTROL
 $\alpha = 0^\circ$; $k_a \approx 0.12$

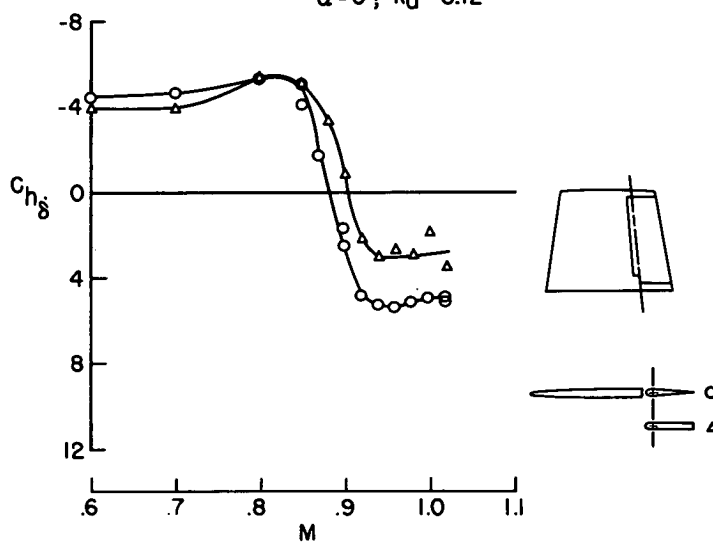


Figure 4

STIFFNESS COEFFICIENTS FOR UNSWEPT-WING CONTROL

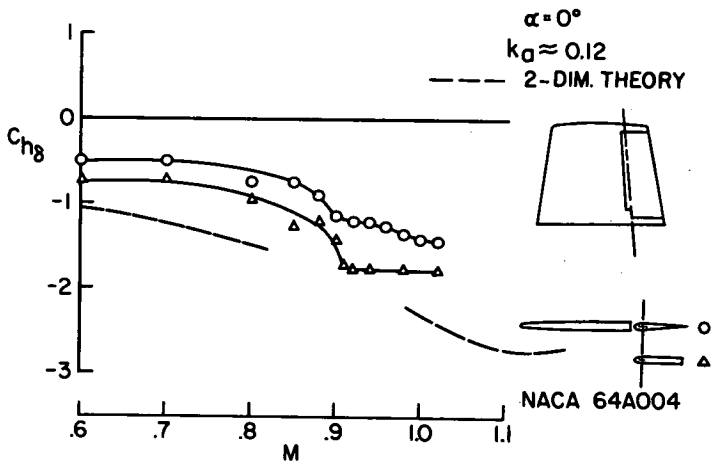


Figure 5

STIFFNESS COEFFICIENTS FOR SWEEPED-RUDDER CONTROL

$k_a = 0.048$; $\delta = \pm 1^\circ$

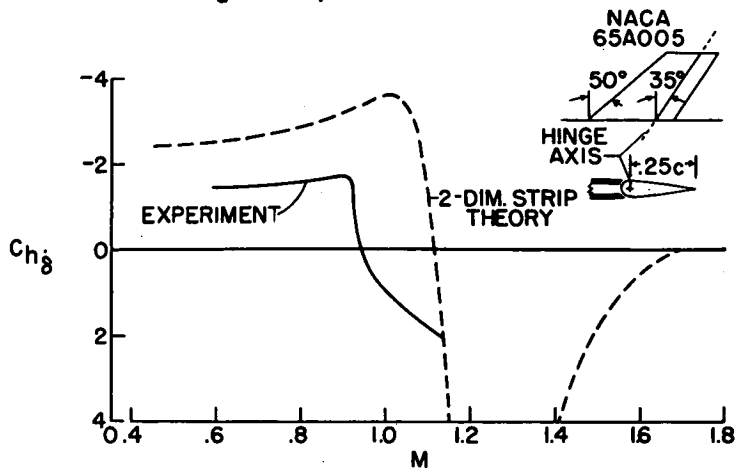


Figure 6

STIFFNESS COEFFICIENTS FOR SWEEPED-RUDDER CONTROL
 $k_a = 0.048; \delta = \pm 1^\circ$

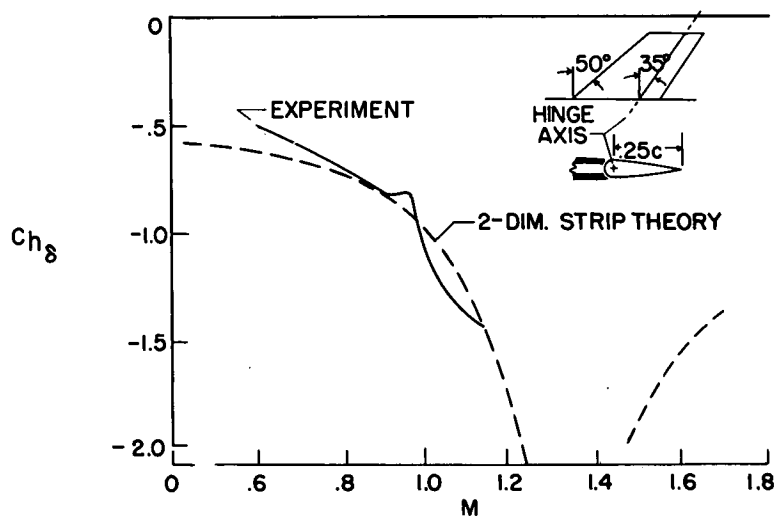


Figure 7

DAMPING COEFFICIENTS FOR DELTA-WING CONTROL

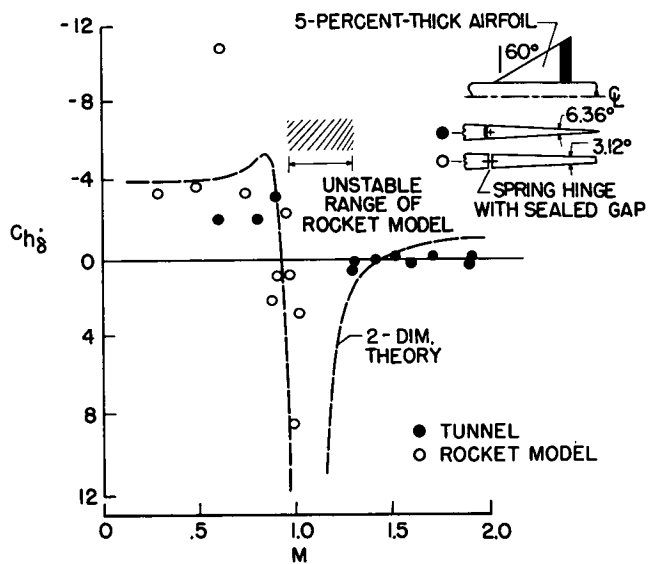


Figure 8

STIFFNESS COEFFICIENTS FOR DELTA-WING CONTROL

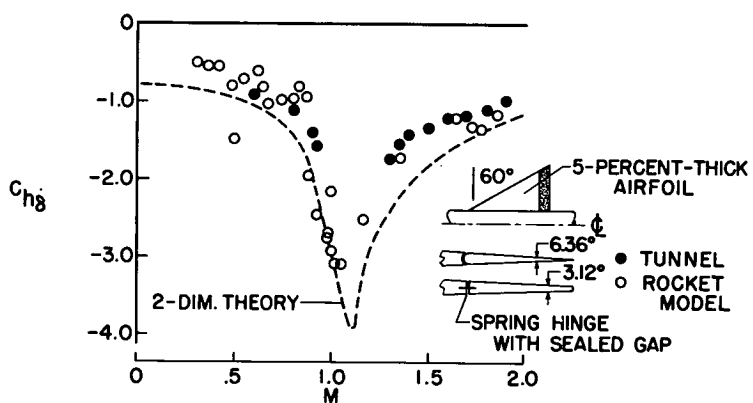


Figure 9